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Reusable Space Transportation Systems

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Major efforts in the U.K. (1984–1994)

2.1 OVERVIEW, OBJECTIVES, AND PROGRAMMATICS

British Aerospace (BAe) and Rolls-Royce (RR) undertook many system and technology studies before the Hotol (Horizontal Take-Off and Landing) concept was revealed to the space transportation community. The project was originated by the Space and Communications Division (Stevenage) of BAe. Between 1982 and 1984 the overall concept was brought together by a Rolls-Royce/BAe team led by John Scott and Bob Parkinson. This concept at that time was completely different from the approach taken in the U.S. (Shuttle) and later in France (Hermes) and Germany (Sänger). It was planned to supplement or even supersede the U.S. Shuttle and the Ariane upper-stage Hermes. It was a single-stage configuration which was launched from a rocket-driven trolley, was completely reusable, and was designed for unmanned operation, primarily for satellite launch and retrieval. The most innovative component of the Hotol concept was the use of a hybrid air-breathing propulsion system, which incorporated the RB 545, a unique liquid air cycle engine (LACE), invented by Alan Bond (later of Reaction Engines Ltd.) and built by Rolls-Royce. In 1985 the project showed great promise and so was matured for a two-year proof-of-concept study (shown in Figure 2.1 [1]).

By 1986 the work was supported by government funding. It was considered an alternative to Hermes, which had been formally defined as an ESA key element since the ESA Council Meeting in Rome (1985). The British delegation's intention was to create an optional ESA program to which British companies could later contribute by providing their pre-

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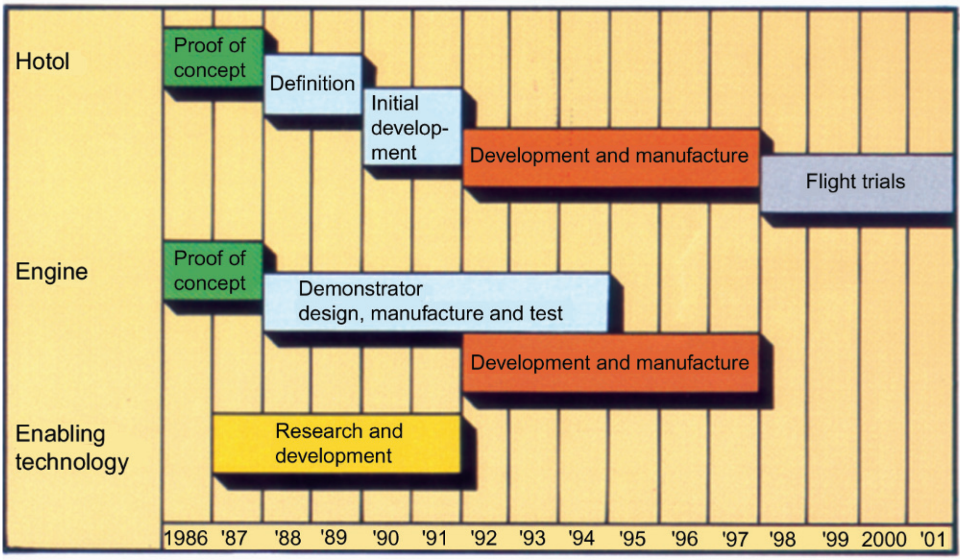

Figure 2.1. Design and development planning for the Hotol concept [1].

financed Hotol activities. At the ESA level the British proposal immediately generated a lot of interest inside the European space community. Unfortunately, in 1988 the British government withdrew further funding but work still continued with industrial support on a lower level.

The motivation underlying further studies was to overcome still unresolved technological problems, specifically regarding aerothermodynamics and operational aspects, such as the launch site necessary for such an SSTO concept. One ESA study made a tradeoff between a pre-cooled turbojet and a ramjet. The ramjet needed an additional rocket to reach its operational ignition speed. It was found that the pre-cooled turbojet propulsion system was inferior to the rocket ramjet propulsion system for a vehicle launch from Europe into an orbit of 28.5°. As a consequence the mission requirements for a new STS had to be defined early. In order to minimize the launch cost, a vehicle like Hotol needed to be launched from an equatorial launch site (e.g., Kourou). If the mission required a launch from Europe then an additional stage would be needed to carry Hotol on top. But a real TSTO needed the design of an additional vehicle. The only possibility to offset this additional cost was to make use of an existing large subsonic cargo aircraft.

In 1990 an agreement was signed between BAe and Russian companies (TsAGI, NPO Molnija, and others) and DB Antonov in the Ukraine to investigate the heavy-lift capacity of the An-225 (Mriya) aircraft which allowed a cargo mass of maximum 250 tons. This meant that Hotol, with its takeoff mass of 567 tons, had to be redesigned (scaled) not to exceed this limit. The result was named Interim Hotol. This vehicle resembled a shortened Hotol. It had four Russian RD-0120 liquid hydrogen/oxygen rockets which were later replaced by the Russian RD-701 tri-propellant rocket engine. This system concept was presented to ESA and the members of the Ariane Program Board in Paris on June 21, 1991. However, this proposal was rejected.

Another project related to Hotol was the Skylon spaceplane proposed by the British rocket scientist Alan Bond in 1989 after the funding for Hotol was terminated by the government. He had analyzed the results from Hotol work specifically regarding the engine concept, the airframe, and the launch mode. Like Hotol, the Skylon concept was an SSTO vehicle, fully reusable, unpiloted, but with more user-friendly operations and an increased cross-range which allowed the vehicle to return to the launch site. Details of the Skylon system concept design were published at the 46th International Astronautical Congress held October 2–6, 1995 in Oslo, Norway.

After termination of Hotol and Interim Hotol, project work turned to the Rocket Ascent Demonstrator Mission (RADEM) study which was conducted between 1993 and 1994. It was financed by ESA and performed by an international team, led by British Aerospace Space Systems Ltd., which included British Aerospace Military Aircraft Ltd. and Russian partners from Molnija, Antonov, and TsAGi. This team had already worked together between 1990 and 1991 on the Interim Hotol project. Therefore, there was a lot of commonality in the approach to the RADEM proposal as a technology demonstrator (e.g., in-flight launch on top of the Antonov 225).

2.2 CONCEPTUAL DESIGN WORK

2.2.1 Hotol

Like all the other studies on new innovative reusable space transportation systems of the 1980s and 1990s in different countries, the driving objective of the Hotol project was reduction of the cost of launching payloads into orbit. This could be achieved by

- horizontal takeoff and landing from conventional runways;
- full reusability of all hardware including a propulsion system with a short turnaround; and
- unmanned operation for transport of payloads to a selected orbit, then retrieving them and returning them back to Earth.

Hotol was the first concept of a fully reusable single-stage-to-orbit vehicle for ascent to low-Earth orbit, proposed by British Aerospace in the U.K. in

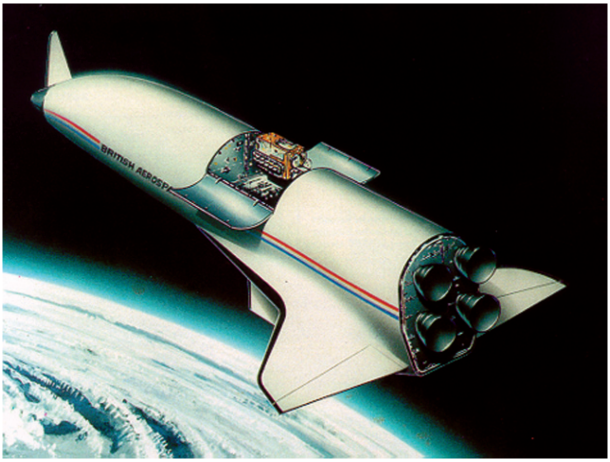

Figure 2.2. Hotol concept from an artist's view [2].

1982. Figure 2.2 [2] shows an artist's view of the concept. It was an unmanned vehicle, designed to take off from a launch trolley which was needed to accelerate the system to a sufficient take-off-speed at which the highly sophisticated airbreathing propulsion system, the Rolls-Royce RB545, provided enough thrust for the climb-out trajectory up to a speed of about Mach 5 at an altitude of nearly 26 km when the hydrogen/oxygen–propelled rocket system took over to reach a low-Earth orbit as the final destination. The technological details of the airbreathing, air-collecting, oxygen-separating, and oxygen-liquefying turbo-based propulsion system was considered top secret for many years until more dedicated propulsion studies and cycle analyses shed light on this completely new propulsion system.

Starting from a baseline configuration (shown in Figure 2.3 [1]), several extensive trade-offs had to be done to achieve the optimum concept for Hotol. The shape of the fuselage—resulting in different drag and structural efficiency (mass/volume)—was systematically investigated. In addition, the rocket and airbreathing propulsion system had to be integrated. The change in shape of the intake—from semi-conical to a 2-D vertical wedge—was one of the most important steps in design optimization of the whole Hotol concept. The final configuration was characterized by the following major changes:

- increased number of engines (from three to four);
- air intake shape changed from conical to 2D vertical wedge;
- removal of the vertical fins;
- removal of the front fuselage canards;
- the fuselage reduced length but increased diameter.

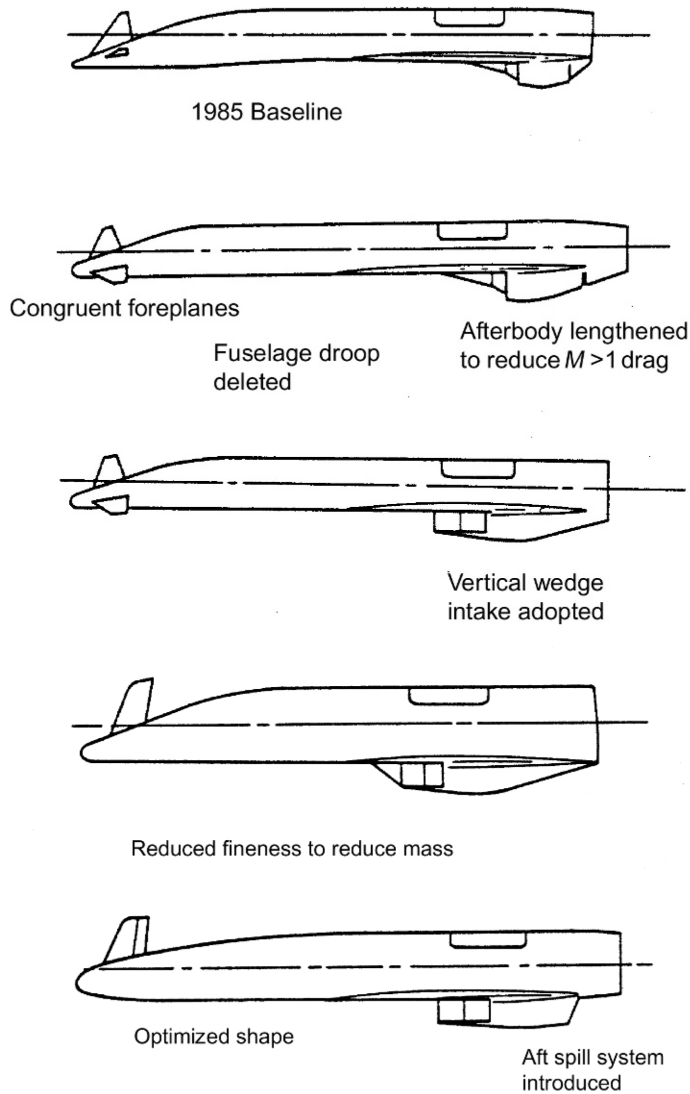

Figure 2.3. Progress in the Hotol system design starting in 1985 (baseline) [1].

All these iterative steps during configuration optimization resulted in important consequences for the design of the structure of the airframe. Heat-resistant material (e.g., carbon-carbon) was planned for the nosecone, the leading edges of the wings and nose fin, intake lips, and central parts of the fuselage's lower surface. Thermal protection system panels were proposed for major parts of the whole fuselage. Figure 2.4 [3] gives an overview of the major structural components of Hotol.

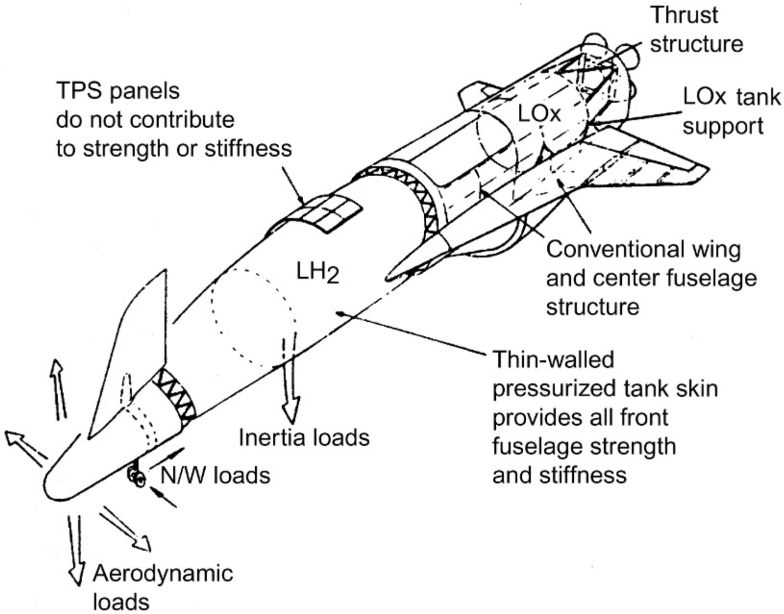

Figure 2.4. Hotol concept structural layout [3].

The final concept was capable of transporting a useful payload of 6.3 tons with a gross takeoff mass of 275 tons (low-Earth orbit). This did not include the undercarriage mass for launching the fully fueled vehicle. This is normally assumed to result in an additional mass of about 3%, which equates to 8.25 tons. But this would not allow for any payload and the undercarriage would have been transported to orbit as a deadweight. For the purposes of landing the empty vehicle, a much less heavy undercarriage system, such as that of a conventional civil transport airplane, would be sufficient.

Two solutions were found to overcome the problems in launching Hotol without a heavy undercarriage. First, there was a trolley which worked as a "launch assist system"—so called by Bob Parkinson. This was an undercarriage system that remained on the ground after liftoff of the launcher. This trolley could be either passive (i.e., without any propulsion system) or active (i.e., with an additional propulsion system). As Figure 2.5 [3] shows schematically, the passive trolley enabled liftoff of the launcher but needed a longer rail for its deceleration. The active trolley led to some fuel saving onboard the launcher for liftoff. Second, there was an air-launched mode that made use of a large transport aircraft (discussed in Section 2.2.2).

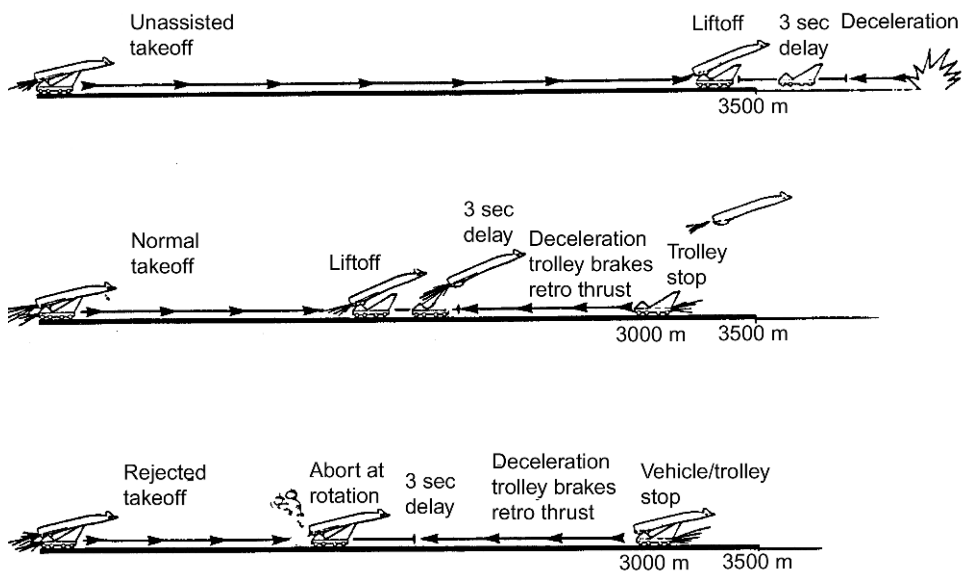

Figure 2.5. Hotol: assisted launch mode using a trolley [3].

2.2.2 Interim Hotol

Using a trolley for launch posed some severe disadvantages. First, the Hotol vehicle would lose a lot of flexibility regarding the launch site. There was only one place where the trolley was available. Any flyback after landing at a different site could not be possible because of the missing infrastructure at different landing sites, even when a trolley was not needed for an empty vehicle. So, the only way to fly back would have been on top of a large transport aircraft like the Russian Antonov 225.

If such a cargo aircraft was capable of carrying back the Hotol vehicle, then maybe it would be possible to launch Hotol from on top such an aircraft at subsonic speed. This procedure would save a lot of investment for the trolley and would solve launch abort problems. On the downside, Hotol would be limited to a gross takeoff weight of no more than 250 tons for the An-225, the largest existing cargo aircraft. Extensive studies started in 1990 of the feasibility of the so-called Interim Hotol being air-launched from on top of an An-225.

Interim Hotol had been a joint study between the Soviet Ministry of Aviation Industry and British Aerospace since September 1990. Participating institutions from the U.K. were British Aerospace Systems

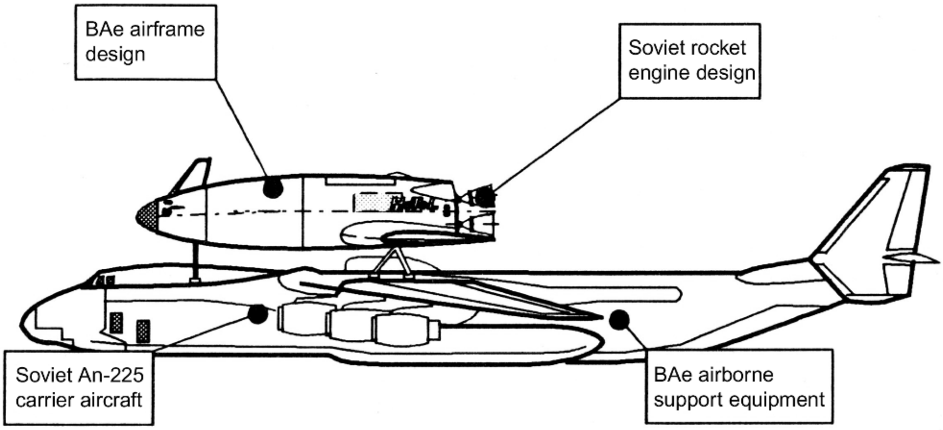

Figure 2.6. Interim Hotol atop the Russian Antonov-225 (MRIJA) [4].

Ltd. (Stevenage) and Military Aircraft Ltd. (Warton) and their counterparts from Russia were TsAGI (Zhoukovski), the Antonov Design Bureau (Kiev), CIAM (Zhoukovski), and the Chemical Automatics Design Bureau (Voronezh). Figure 2.6 [4] shows the basic work share between British and Russian partners:

- Russia provided the carrier aircraft (An-225) and the Soviet RD-0120 hydrogen/oxygen rocket engines which should be reusable for between 20 and 25 launches;
- British Aerospace was responsible for the orbiter airframe design and some airborne support equipment for the An-225.

The objective of the studies was "to examine the possibility of designing an economically attractive, fully reusable, rocket-powered launch vehicle using an airborne launch from the An-225 Heavy Lift Carrier Aircraft."

The major outcome of these studies was presented to ESA in the summer of 1991 and then to the AIAA/DGLR Fifth International Aerospace Planes and Hypersonics Technologies Conference, held in December 1993 in Munich, Germany. Figure 2.7 [5] summarizes the most important configuration changes from Hotol to Interim Hotol as follows:

- four Russian engines (RD-0120) with dual-expansion-ratio bell nozzles;
- reduced body fineness ratio (resulting in reduced drag);
- removal of the intake and addition of skin panels at the intake location;
- smaller wing and shortened LH_2 tank;
- removal of heat exchangers and turbo-compressors;
- aft fin location in place of a nose fin (realized in a second design cycle).

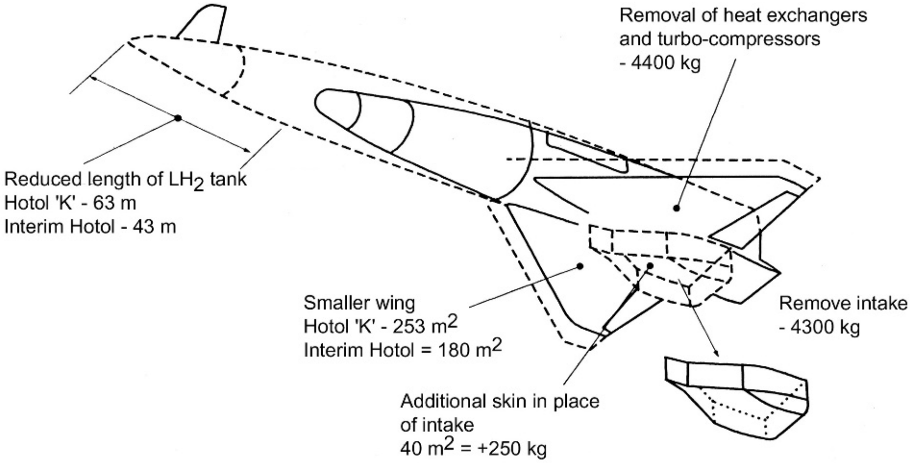

Figure 2.7. From Hotol to Interim Hotol: configuration changes [5].

The concept is shown in an artist's view after separation of the orbiter from the An-225 in Figure 2.8 [5]. Separation was intended to take place at Mach 0.8 at an altitude of 9 km. The separation procedure is shown schematically in Figure 2.9 [4]:

- turn onto heading and set up launch (at $T - 10\,\text{min}$, $H = 9.4\,\text{km}$);
- powered dive (at $T - 62\,\text{s}$, $H = 9.4\,\text{km}$);

Figure 2.8. Interim Hotol and An-225 after separation [5].

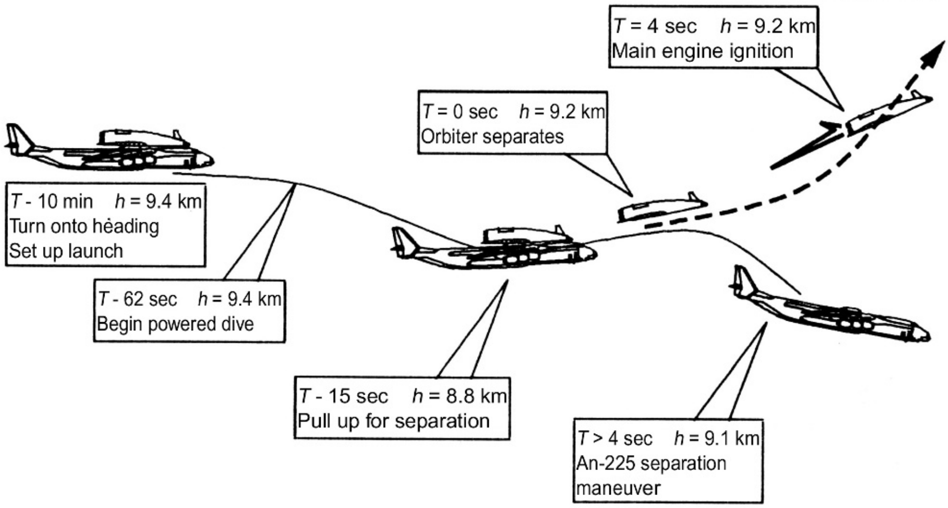

Figure 2.9. Interim Hotol: separation from An-225 [4].

- pull up for separation (at $T - 15\,\text{s}$, $H = 8.8\,\text{km}$);
- orbiter separates (at $T = 0\,\text{s}$, $H = 9.2\,\text{km}$);
- the An-225 carries out a separation maneuver ($T > 4\,\text{s}$, $H = 9.1\,\text{km}$);
- main engine ignition ($T + 4\,\text{s}$, $H = 9.2\,\text{km}$).

The most important requirement in the design of the orbiter was the maximum carrier load of the An-225 (i.e., 250 tons). Finally, as a result of the first iterative design effort, the mass breakdown for the Interim Hotol stayed within this mass limitation as shown in tons in the following table:

Inert vehicle	*Tons*
Body	19.1–17.3
Wing	3.2–3.1
Equipment	3.1–2.8
Propulsion	6.5–5.9
Propellant	
LH_2	29.6
LOx	177.4
Auxiliary propulsion	1.7
Residuals	2.0
Payload	5.1–7.9
Contingency margin	2.5
Total	*250*

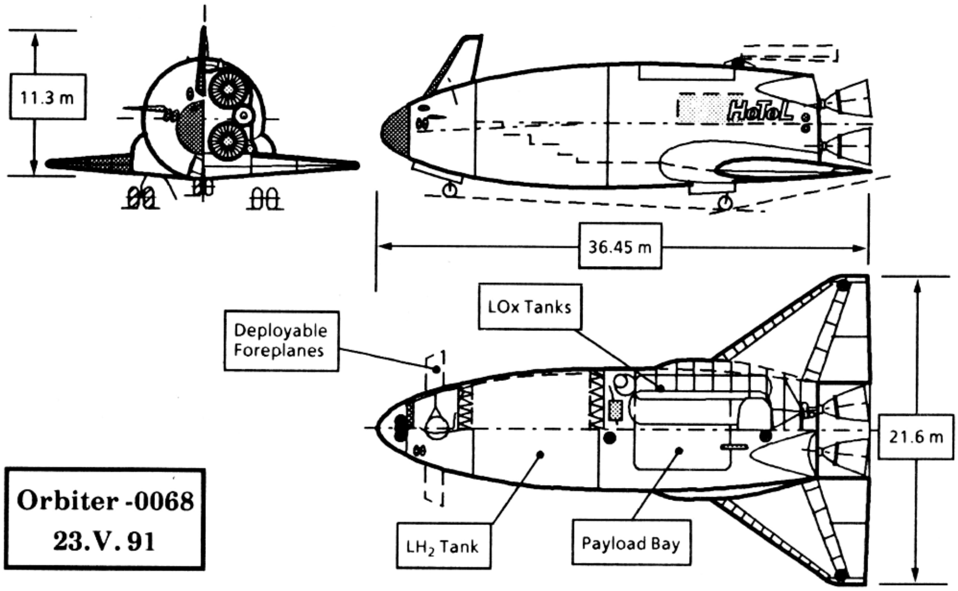

Figure 2.10. Interim Hotol: major dimensions [4].

The table shows the range in estimate uncertainty regarding materials and structure. The geometrical dimensions of this first design step are shown in Figure 2.10 [4]. These results were the baseline for continued work which resulted in a much more detailed structural design. As used for Hotol, the nose fin and the deployable fore-planes were still present. But during in-depth aerodynamic stability and control performance calculations, it turned out that these controls were no longer needed. This was due to the decreased fuselage fineness ratio and moving the nose vertical fin to the rear fuselage. Finally the rocket propulsion concept based on four RD-0120 engines was changed to the RD-701 tri-propellant engine being developed by the Russian NPO Energomash. This engine used kerosene in addition to liquid hydrogen. It increased the average density of the propellants which resulted in a reduction in tank volume, then fuselage volume, and finally structural mass. This final design, shown in Figure 2.11 [6], allowed a payload increase of about 1.5 tons.

Compared with the original Hotol concept, Interim Hotol represented a much lighter and technologically less risky approach to carrying the same payload into orbit. In the early 1990s nearly all system components for the Interim Hotol related to propulsion and the carrier launch vehicle, both of which were ready and available. This had to result in a much lower systems development cost. Remaining uncertainties about the need for aerothermodynamic heatloads and, consequently, heat-resistant materials for the re-

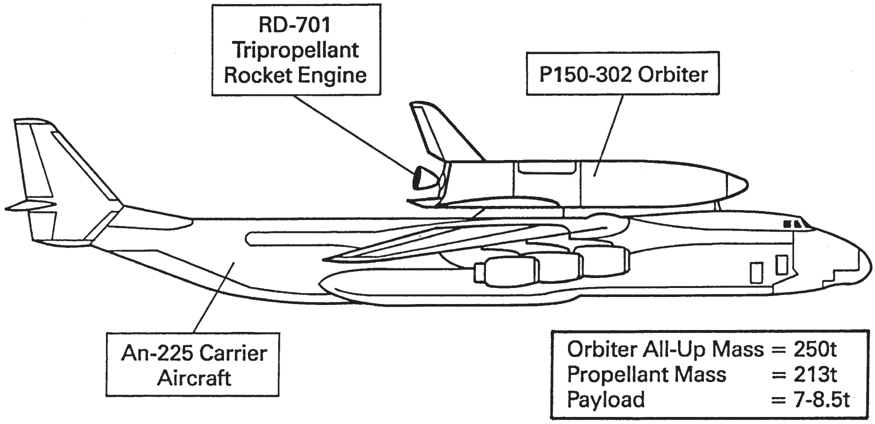

Figure 2.11. Interim Hotol: An-225 and orbiter with tri-propellant rocket engine. Courtesy of British Aerospace [6].

entry procedure could also be readily solved in Europe from Hermes and Ariane project experience (e.g., the development and tests of shingles for the TPS and the use of CFRP sandwich skin panel structures developed for Ariane V's upper stage).

2.2.3 Skylon

After the Hotol project was brought to an end—as a result of the British government's refusal to invest further money in 1988—and the Interim Hotol was ended in 1991, the RB545 propulsion system remained thereafter subject to a patent security classification. This prevented the British companies (i.e., BAe and RR) from discussing the projects with other potential European partners. At that time members of the Rolls Royce propulsion team founded a new independent company called Reaction Engines Ltd. This company designed a completely new synergetic airbreathing rocket engine (SABRE) as the basis of a new space transportation vehicle, called Skylon, which should overcome the technical deficiencies (e.g., sled launch) encountered during development of the original Hotol. Specifically, the engineering experience gained during Hotol structural design work was implemented. The result was an increased payload of 12 tons into low-Earth Orbit (LEO) for the same takeoff mass of 275 tons. After cancellation of the U.S. aerospace plane (NASP), Skylon was considered worldwide as the first SSTO. It was an aircraft that used atmospheric oxygen in its rocket mode during climb (up to about Mach 5 and 26 km altitude) and pure rocket technology (LOx/LH_2) when it left the atmosphere for LEO. This SABRE engine concept was attractive enough for ESA to finance some limited studies to provide infor-

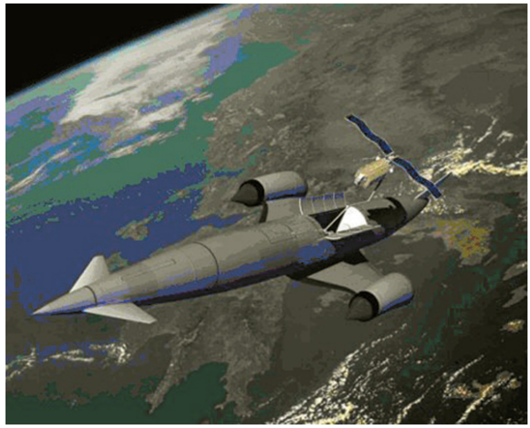

Figure 2.12. Skylon: in flight artist's impression [7].

mation on engine performance. Figure 2.12 [7] is an artist's impression of Skylon delivering a payload.

Skylon was designed to take off and land using a relatively conventional retractable undercarriage. The fuselage was 6.5 m in diameter, 82 m in total length, and had a wingspan of 25 m. The major features of the Skylon system can be summarized as

- reusability—reduced cost per flight by amortizing the production cost over up to 200 missions;
- SSTO—a single-stage concept was superior to multi-stage vehicles because they were less costly to develop and had lower operational cost;
- unpiloted—onboard computers provided flight control (unpiloted flight reduced the vehicle development cost because required safety standards could be relaxed);
- abort capability—the vehicle was capable of flying safely to contingency landing sites even with an engine shut down;
- user-friendly operation—a tractor could tow the vehicle on its own undercarriage and there was minimal maintenance in between flights as a result of robust TPS and reliable engines;
- re-entry cross-range—the low re-entry wing loading resulted in a favorable hypersonic L/D and the vehicle could return to the launch site even from a highly inclined orbit;
- environmental impact—the SABRE engines used environmentally friendly propellants and the system did not contribute to the space debris problem.

The Skylon configuration is shown in Figure 2.13 [8]. In contrast with other reusable space transportation systems (e.g., NASP and the Sänger first stage),

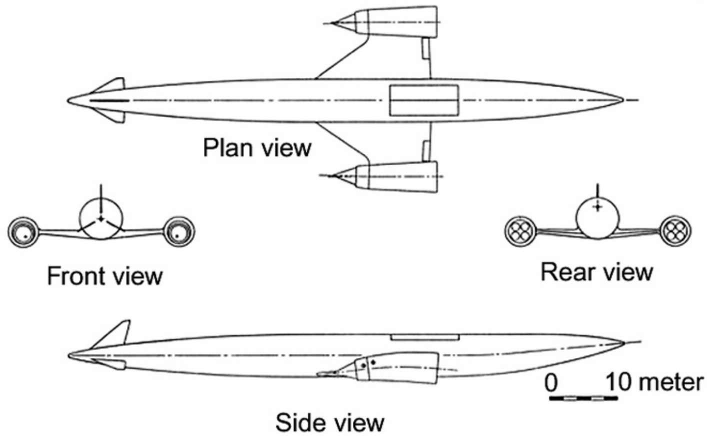

Figure 2.13. Skylon: Configuration A4 (1991) [8].

Skylon did not have a fully integrated airbreathing propulsion system. The wings, the tip-mounted engines, the payload bay, and the location of the heavy LOx tank led together to a center of gravity (CoG) of about 55%. A foreward LH_2 tank and a separate rearward LH_2 tank completed the balanced tankage design shown in Figure 2.14 [8].

With conventional vertical rocket designs, heavy engines were located at the rear of a blunt-based fuselage. Since the dry CoG is dominated by the engine location, the wings and fueled CoG (i.e., the LOx tank) also had to be in the rear. Therefore, the payload bay and the LH_2 tanks had to be integrated in a large fore-body. These configurations suffered from a large center of pressure (CP) and CoG mismatch during flight in the atmosphere. The CP

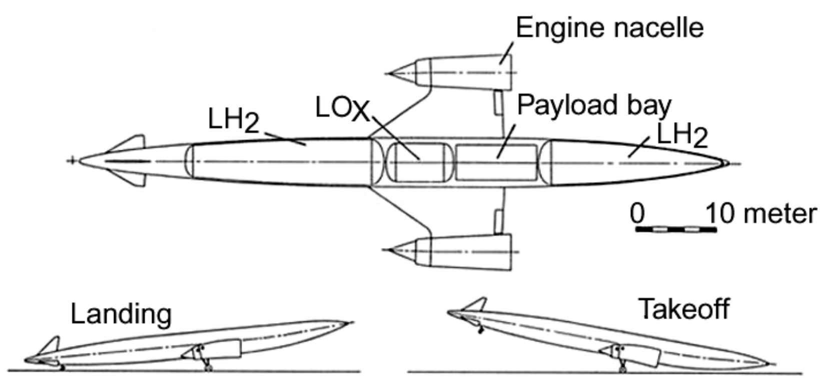

Figure 2.14. Skylon: internal layout, tank locations [8].

shifted 10 m forward partly as a result of the wide Mach number range and partly as a result of the large forward fuselage cross-section area. During the Hotol project, this resulted in critical trim problems that finally lowered the usable payload. The Skylon design, however, moved the CoG from about 75% to about 55% resulting in stable control during flight in the atmosphere by fore-planes in pitch, wing ailerons in roll, and an aft-mounted fin in yaw. As for the Hotol and Interim Hotol, a nose fin was originally planned but later dropped. The Skylon configuration, which evolved from a design review of the Hotol airframe, represented an efficient resolution of the problems encountered there.

Fuselage aeroshell

The aeroshell formed the outer surface of the aeroplane and must withstand local aerodynamic pressure loads and kinetic heating. For spaceplanes three main structural possibilities were proposed:

- a hot structure using reinforced ceramics;
- a cold structure where the main propellant tanks had to be permanently pressurized so that it could handle the maximum fuselage-bending moments without buckling;
- introduction of a separate tailored structure from which the tanks and aeroshell were suspended.

Figure 2.15 [9] shows the third option, which was chosen for the Skylon fuselage. The complete fuselage insulation system was layered consisting of multi-foil blanket thermal protection insulation (TPI), an airgap (suspended by a cylindrical fuselage girder made from a carbon fiber reinforced product or CFRP), and finally the tank foam insulation. During re-entry the maximum surface temperature is kept down to 1,100 K by dynamically controlling the trajectory. The aeroshell temperature was then 800 K hotter than the CFRP substructure.

SABRE engine

Reaction Engines Ltd.'s synergetic airbreathing rocket engine (SABRE) was a hybrid engine which used air up to about Mach 5 and 26 km altitude. Onboard, liquid oxygen was employed beyond this speed and altitude, when the engine operated as a closed-cycle rocket. A variable axisymmetric intake supplied air to a radial flow precooler and a high-capacity bypass duct (see Figure 2.16 [8]). The precooled air was compressed to high pressure and fed into a pair of twin-chambered rocket engines. The engine cycle was

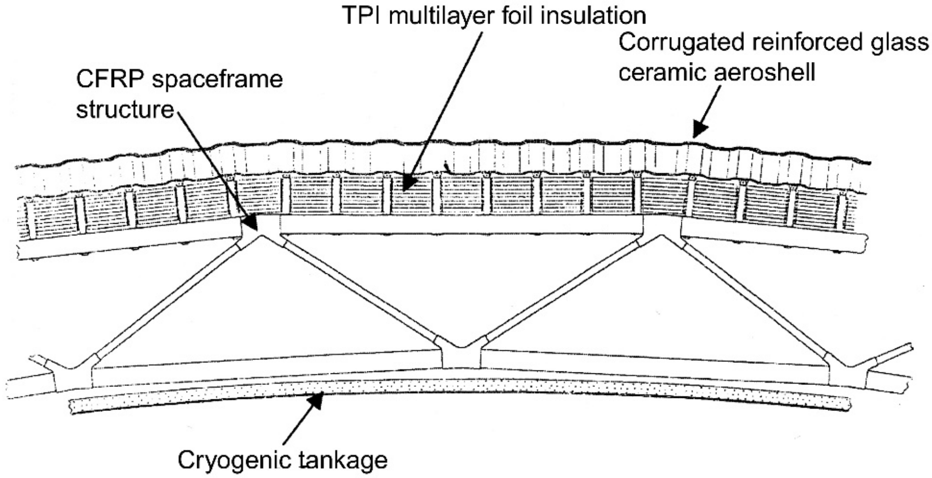

Figure 2.15. Skylon: structural concept for the fuselage [9] (CFRP = carbon-fiber-reinforced product; TPI = thermal protection insulation).

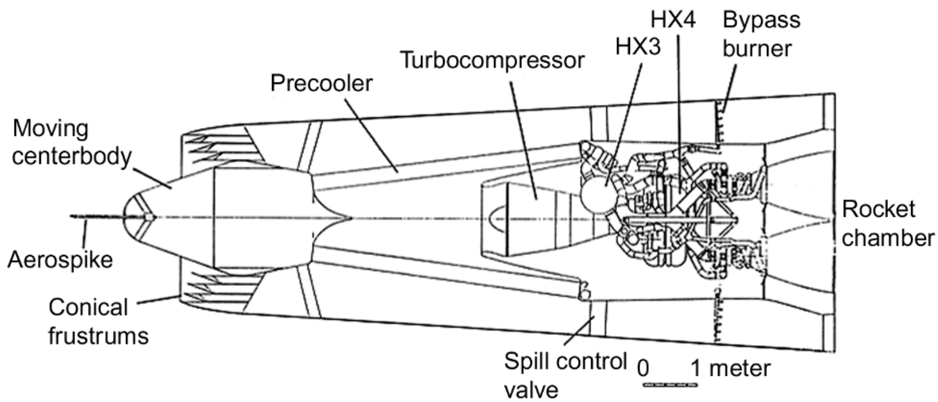

Figure 2.16. Skylon: SABRE engine [8].

over-stoichiometric as a result of its thermodynamics. Some of the excess hydrogen was employed in a bypass duct burner to reduce spill drag at the intake.

Since the airbreathing mode operated on a turbo-machinery–based cycle, the engine is capable of generating static thrust (unlike ramjet engines), and engine-testing and development can therefore take place in conventional open test facilities. The SABRE engine was designed with state-of-the-art technology for turbo-machinery, pumps, combustion chamber, etc. Current materials were specified for the engine machinery while the nacelle shell could be manufactured in SiC-reinforced glass and the bypass system in

C-SiC. By employing a rocket combustion chamber, nozzle, and pumps in both modes, the mass penalty of adding a separate airbreathing engine could be reduced.

The SABRE engine cycle involved a hot airstream (coming from the intake) and a cold hydrogen stream (coming from the liquid hydrogen tanks). For cooling down the airstream and warming up the hydrogen stream a working fluid that remained gaseous was needed. Helium was chosen. Most SABRE engine components (e.g., combustion chambers, nozzle, pumps, and turbo-compressor) were relatively conventional. Lightweight high-power heat exchangers, however, were a new feature peculiar to this type of engine. This turned out to be the most challenging manufacturing problem which has yet to be solved in practice even today.

In addition, installation of the engine at the wing tips posed some engineering problems. The intake entry was positioned ahead of the wing in close free-stream conditions to prevent wing or fuselage ingestion and wing shock interaction. In addition, the intake axis was dropped 7° nose down relative to the fuselage centerline to minimize the effects of vehicle flight angle incidence. This can be seen by comparing Figures 2.13 and 2.16.

Skylon mass breakdown

The mass estimate (as at 1995) for the Skylon Configuration C1 is listed in the table at the top of the next page.

Summarizing remarks

The material given in this chapter was first presented to ESA in 1991 and, second, after more detailed further development studies, at the International Astronautical Congress held 1995 in Oslo. The work on Skylon was partly based on Hotol investigations. This was especially true for areas like re-entry heating, TPS, thermal control, and materials. But some of the deficiencies within the Hotol project had been resolved by a new vehicle configuration.

Major conclusions about the Skylon design can be summed up by the following statements:

- an SSTO spaceplane operating from Kourou could deliver up to around 12 tons into an equatorial low-Earth orbit with a total gross takeoff mass (GTOM) of 275 tons;
- such an SSTO was feasible using a new advanced hybrid airbreathing rocket engine employing subsonic combustion and based mostly on hardware concepts applied at that time in rockets, gas turbines, and ramjets;
- materials technology for such an SSTO was constantly improving, in particular C-SiC (carbon-silicon/carbide) for TPS and hot engine

	Mass (kg)	
Main engines	9,628	
Nacelle, inlet, bypass	3,500	
Wing	5,115	
Fuselage: aeroshell, insulation, structure, payload bay	8,130	
Main tanks, cryo-insulation	2,816	
Undercarriage	4,170	
Aerodynamic control, hydraulics	2,660	
Auxiliary systems, pressurants, coolants, etc.	5,016	
Basic mass		*41,035*
OMS/RCS propellant	2,357	
Ascent fuel	66,807	
Ascent oxidizer	150,235	
Propellant margins and residuals	1,282	
Total fluids		*220,681*
Mass margin		1,284
Payload		12,000
Gross takeoff mass		*275,000*

components, carbon/peek for tanks and structure, reinforced titanium and nickel alloys for wings and aeroshell, and carbon-reinforced aluminum for engine components;

- selection of the correct aerodynamic configuration resulted in a trimmable vehicle without excessive fluid transfer complexity, induced drag, or mass penalties;
- sufficient performance margins for structural mass estimates were included in the mass breakdown.

2.2.4 Radem

Interim Hotol studies had already shown the attractiveness of air-launching a new space transportation system. A vehicle which could be air-launched had some specific advantages inherently associated with this operational mode:

- use of a higher expansion ratio for the rocket engine than would be possible for a ground-launched vehicle;
- reduction in drag losses, allowing a higher terminal Mach number to be achieved with a smaller vehicle;
- ability to launch at a point distant from the home base and to return home at the end of the mission.

But, there is no such thing as a free lunch. The requirement of a (large) carrier aircraft resulted in additional cost. However, this amount of money was known and could be specified precisely. A second concern for air-launching a highly sensitive and costly payload is safety. This meant the risk involved in

- safe vehicle separation;
- the potential need to dump fuel;
- engine ignition near the carrier aircraft.

The Russian An-225 was capable of carrying loads up to 250 tons. It provided flexibility regarding the launch site and provided the cross-range to fly a reusable STS back to a home base in Europe. When studying the feasibility of these new reusable launcher systems the design engineers in different European countries commonly reached the conclusion that there needed to be advances in major key technologies in order to achieve a reasonable payload that satisfied the minimum gross takeoff mass of the system. There was a big question mark about the feasibility of reusable rocket engines, heat-resistant materials and structures, and, last but not least, guidance navigation and control (GNC). All these technologies had to be developed, built, and tested in a realistic flight environment.

The overall governing motivation for Radem studies can be summarized as

- investigating the feasibility of constructing an air-launched, reusable demonstrator vehicle as a means of proving the technology associated with future low-cost space transportation systems at an early date without committing to full-scale development of such a system;

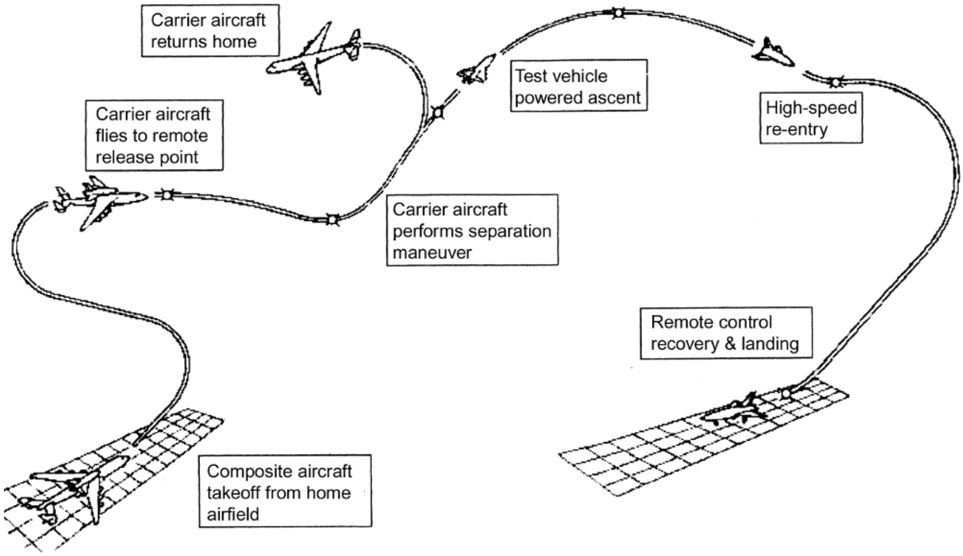

Figure 2.17. Radem: basic concept for an air-launched demonstrator [10].

- evaluation of the advantages of making such a program a collaborative venture between ESA and Russia/Ukraine.

Technology readiness level TRL-6 had to be demonstrated before any technology could be applied to an advanced new launcher system. There was only one way to do it. Early in the 1990s the U.S. had proven the feasibility of flying an experimental rocket-propelled aircraft—the X-15—at hypersonic speeds using a B-52 subsonic aircraft to launch it in-flight. The Russians also used their experimental BOR vehicles to demonstrate the feasibility of technologies planned for the Buran orbiter. This scenario was studied by ESA member states who reached the conclusion that the feasibility of developing and flight-testing an air-launched reusable rocket-propelled demonstrator vehicle needed to be studied to find the key technologies which were mandatory for the Interim Hotol. For such purposes a rocket ascent demonstrator mission (Radem) was defined and the Russian An-225 was again the carrier aircraft of choice. If a small-scale flying testbed could be flown successfully at an affordable low cost the results could constitute a database for building a full-scale STS like Interim Hotol.

Four typical missions were planned for Radem:

- normal flight demonstration (see Figure 2.17 [10]);
- aborted flight demonstration;

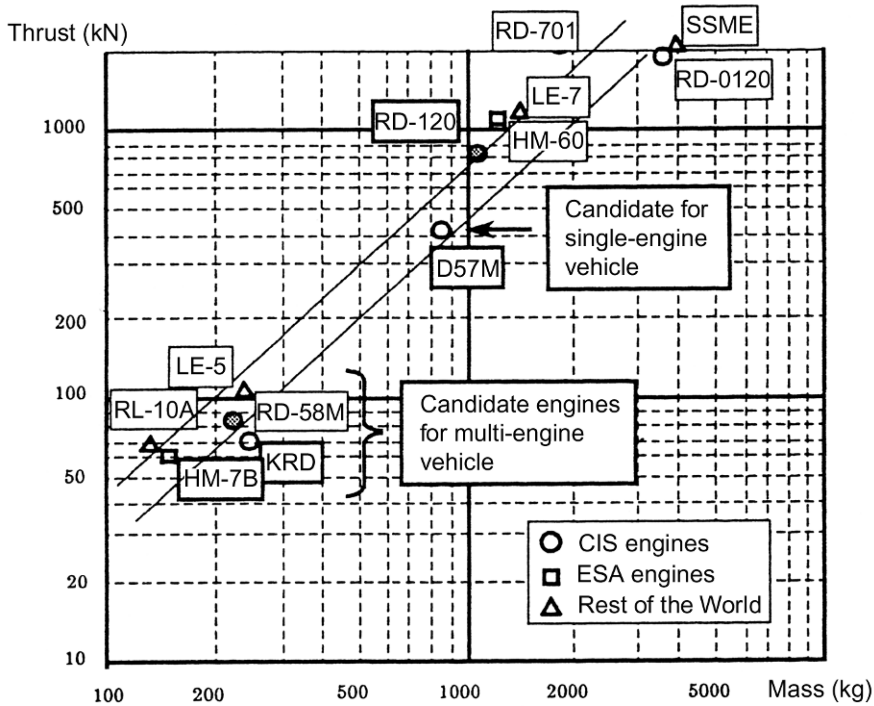

Figure 2.18. Radem: candidates for rocket propulsion demonstrator concepts [11].

- calculating constant dynamic pressure trajectories as part of the experimental program;
- microgravity flights as a potential commercial benefit.

The team who were involved in the Interim Hotol study were a logical choice to carry out the study work. Radem studies were conducted between February 1993 and January 1994. As for Interim Hotol, the study team was led by British Aerospace Systems Ltd. and included NPO Molnija, the Antonov Design Bureau, TsAGI (the Central Aerodynamics and Hydrodynamics Institute at Zhukowsky), and British Aerospace Military Aircraft Ltd.

The most important decision for the Radem configuration was selection of the rocket propulsion system. Figure 2.18 [11] shows a compilation of all existing candidate rocket systems. At the time there was a group of engines with a thrust of about 70 kN (which led to a small vehicle) and a second group with a thrust of about 1,000 kN (which led to a large vehicle). In addition to liquid oxygen/hydrogen–fueled engines, liquid oxygen/kerosene engines were included in the study work (which led to a tri-propellant propulsion system). A European engine was to be given preference against existing non-European ones. Because the Vulcain engine was about twice as powerful as a small Radem vehicle, a cluster of HM-7B engines was selected.

Comparison of Radem engine configurations:

6 × HM-7B (Radem 1603)	*RD-120 + 2 × HM-7B (Radem 2802)*
All LH_2/LOx propulsion	LOx/kerosene boost + LH_2/LOx sustain
• Provides liquid hydrogen experience in a reusable vehicle • Direct demonstration of integral tank technology • Simpler propellant supply system • Very large number of rocket engines • Very large volume of hydrogen tank makes the problems of stability and controllability more difficult, and results in a lower lift-to-drag ratio	• Provides liquid hydrogen experience in a reusable vehicle • Increases propellant mass without increasing vehicle size • Provides a higher maximum Mach number for a given vehicle • Reduces the number of engines required • Mixes CIS and ESA rocket engines (taking both industries into account) • More complex propellant supply system results in a lower lift-to-drag ratio

A second area where key technologies for Radem would be needed was the design of the reusable cryogenic tanks. Aluminum–lithium was chosen as a suitable material and the Russian partners in the team already had practical experience in building and testing tank demonstrations. The question of whether the tank should be an integral ("conformal") structural part of the vehicle fuselage led to two potential structural concepts for the vehicle. The first (series 1xxx) adopted an integral tank (similar to that proposed for Interim Hotol). The second (series 2xxx) used a separate airframe and an installed (non-conformal) tank which had already been developed in Russia for a similar concept. Figure 2.19 [11] shows the performance of the two major families for the vehicle concept.

Finally, tradeoffs led to the selection of two vehicle configurations worthy of more detailed design. One design (named "1603") used HM-7B engines only and a conformal integral tank for the liquid hydrogen fuel (LH_2). The second design (named "2802") had in addition to two HM-7B engines a Russian RD-120 liquid LOx/kerosene engine and a non-conformal installed LH_2 tank. Both the finally selected concepts could fulfill all the priority

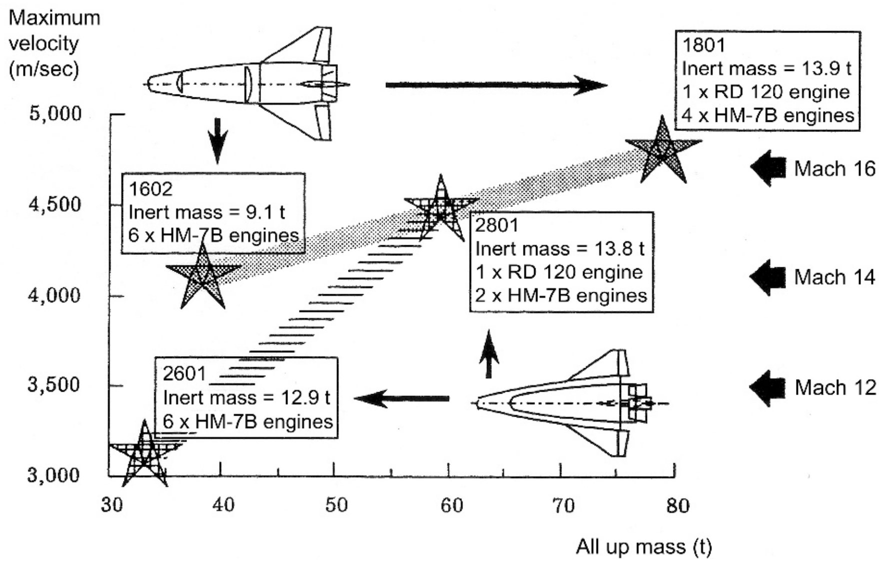

Figure 2.19. Radem design variants for air-launched rocket demonstrator [11].

technical objectives of Radem which were

- to gain experience in the construction of a reusable air-launched cryogenic rocket vehicle;
- to demonstrate in-flight separation and ascent control including validation of the control algorithms;
- to gain experience in the in-flight management of cryogens;
- to obtain aerothermodynamic data and demonstrate TPS and structural concepts;
- to gain experience in ground-handling operations.

Some secondary technical objectives were also defined, which were

- to test materials and structures for hypersonic vehicles;
- to test engine components for airbreathing hypersonic engines needed in other European programs such as ramjet (e.g., for Sänger), air-cooled cycle engines (e.g., for Skylon), and ramjet/scramjet (e.g., military applications);
- to validate hypersonic aerodynamic flight prediction codes;
- to provide limited duration microgravity flight opportunities.

Figure 2.20 [12] shows the major components of finally selected Radem system concepts. Mass breakdown and performance for the two designs is shown in the following table.

	Version 1603	*Version 2802*
Structural parts		
Structure	4,610 kg	4,040 kg
Installed tanks	570 kg	1,270 kg
TPS	1,140 kg	1,663 kg
Equipment		
Electrical	442 kg	442 kg
Hydraulics	359 kg	11,359 kg
Undercarriage	385 kg	680 kg
RCS	360 kg	360 kg
Propulsion system	1,080 kg	1,235 kg
Others	172 kg	172 kg
Main engines	1,100 kg	1,730 kg
Mass margin	722 kg	752 kg
Dry mass	***10,940 kg***	***12,002 kg***
Fluids		
LH_2	3,800 kg	900 kg
LOx	18,200 kg	33,700 kg
Kerosene		11,300 kg
Other	560 kg	811 kg
GTOM	33,500 kg	58,713 kg
Maximum Mach number	12.3	15.7

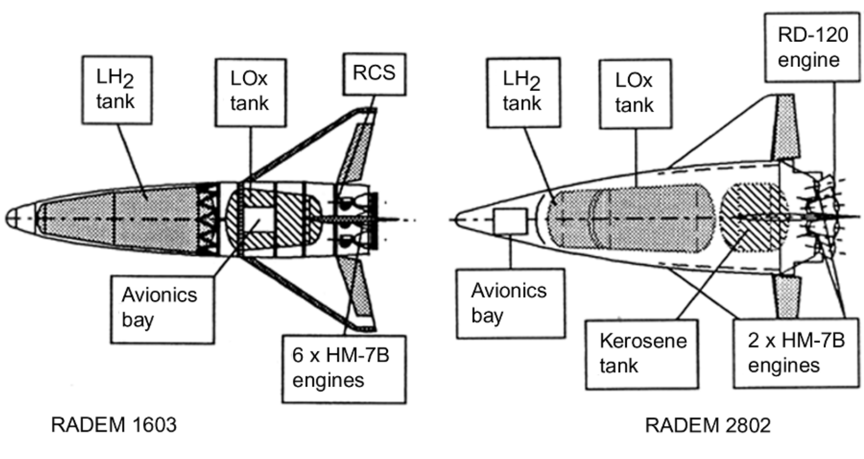

Figure 2.20. Radem: selected candidates for air-launched rocket demonstrator [12] (RCS = reaction control system).

The final comparison of selected Radem configurations

RADEM 1603 configuration	*RADEM 2802 configuration*
Axisymmetric conical body with arrow wing and integral hydrogen tank	Lifting body with delta wing
• Lower structural mass • Direct demonstration of integral tank technology • Possibility of optimizing configuration • Poorer stability characteristics and lower L/D • Increased technical risk associated with structure and application of thermal protection system results in a lower lift-to-drag ratio	• Higher L/D ratio • Good stability and controllability characteristics • Fewer TPS and cryogenic insulation problems due to separation of tanks from the skin structure • Extensive experimental and theoretical background on structure and aerodynamics for a similar Russian system (MAKS) • Possibility of direct usage of structure and (some) equipment in a follow-on operational vehicle • More complex configuration • Higher mass of structure

The An-225 was capable of taking off from a conventional runway and flying the Radem vehicle to a release point approximately 700 km distant and at an altitude of 9,000 m and returning to the launch site. Takeoff mass would be about 382 tons for the 1603 vehicle and 405 tons for the 2802, requiring a takeoff runway of 1,500 m at 15°C and a landing distance of about 1,750 m.

As already mentioned, one of the most critical issues of air-launching a vehicle from the upper side of a carrier airplane was the separation process. Both Radem configurations had a lower lift capacity in comparison with the An-225. Before breaking the mechanical connections, the Radem + An-225 composite should perform a pull-up maneuver in the vertical plane to achieve an optimum separation angle and then carry out a negative g maneuver. The normal g-load would then be decreased to $-0.8g$ for separation. After breaking the mechanical connection, the angle of attack of Radem increased to $+14^\circ$ and the An-225 continued moving with a negative normal g-load of about -1. Separation strategies in which all engines were ignited before separation or only two engines ignited before separation (at about -2 s) were investigated. With only two engines ignited before separation (and the

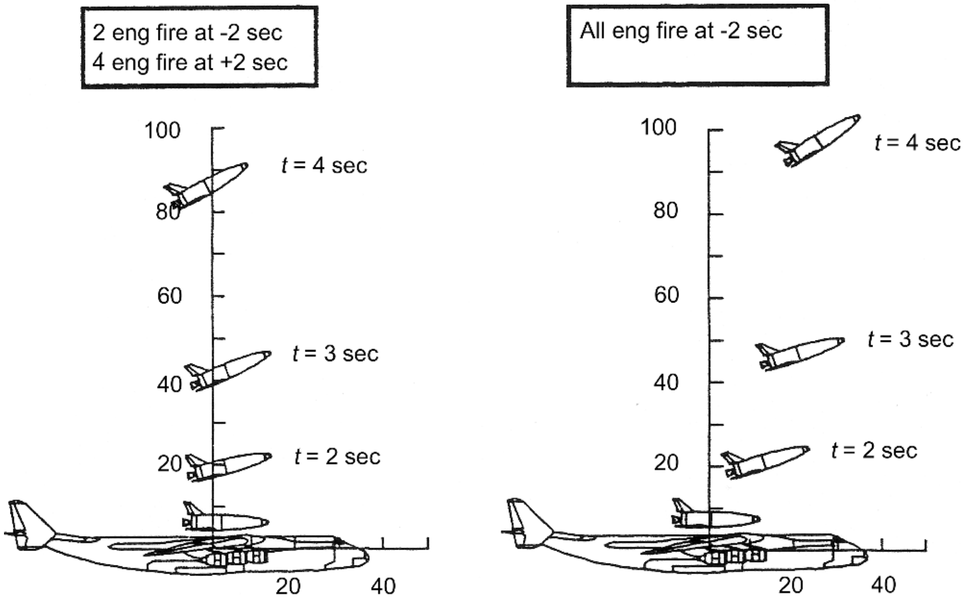

Figure 2.21. Radem separation process [11].

remainder igniting at +2 s) the Radem vehicle 1603 separated on a more or less vertical path with respect to the An-225. With all engines ignited, both configurations of the Radem vehicle moved forward of the carrier aircraft. These separation trajectories are shown in Figure 2.21 [11]. Some additional calculations were made to assess the surface temperature at the rear of the An-225 during separation as a result of plume impingement of the Radem vehicle's rocket engines.

Development program

The approach taken to structuring the development program for Radem was to provide for progressive integration and risk reduction through an aircraft-like program of ground-testing and then progressive flight-testing. The following major ground test rigs were identified:

- structure/thermal test model;
- propellant system test assembly (tanks + propellant management);
- RCS test rig;
- flight control software simulator;
- "iron bird" systems test rig.

Development of all systems was not required for the first test flight. Initial flights were planned to be made with the Radem vehicle without its propulsion system captive on its An-225 carrier. For this unpowered Radem vehicle, the center of gravity had to be kept the same to correctly simulate aerodynamic stability and control. After the pull-up/pull-down maneuver of the AN-225, the free-flying Radem should undertake unpowered flight, gliding to a landing site. The rocket engines would then be required to carry out progressive powered flight trials, gradually increasing the maximum speed reached. Before hypersonic flight was attempted, the vehicle did not have the requisite thermal protection, nor an RCS system. It is possible that the first flight vehicle could lack these systems to reduce initial costs. A second vehicle, introduced about one year downstream from the first, could then include these systems, extending the flight regime to maximum velocity, thermal loads, and range trials.

The likely development schedule of the program is shown in Figure 2.22 [12]. This indicates a 4-year program between initiation of configuration definition and full-scale development, and the first powered flight, with most development test rigs appearing in Year 2. The schedules shown are consistent with the program carried out by North American on its X-15 program, and with the British Aerospace Experimental Aircraft Program (EAP).

The operational approach taken to carry out its role as a demonstrator required the Radem vehicle to be easy to maintain and have a reasonably rapid turnaround time between testflights. The key features in keeping the turnaround time reasonably rapid and yet requiring relatively low levels of

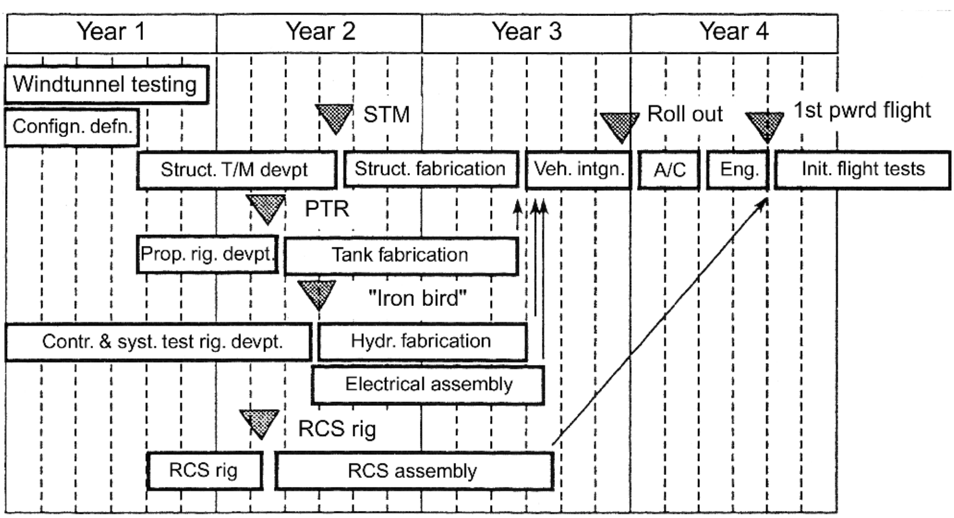

Figure 2.22. Radem: proposed development program [12] (PTR = propulsion test rig; RCS = reaction control system; STM = structural and thermal model).

effort were

- performing maintenance activities with residual (warm) hydrogen in the hydrogen tank;
- using the engine a set number of times without detailed inspection between each flight;
- reducing the time required for post-flight surface inspection;
- minimizing the difficulties in dealing with residual hydrazine in the RCS system between flights;
- providing ready access to subsystems requiring maintenance, particularly in the avionics area.

It appeared reasonable at the time to aim for a full turnaround in about 88 shift-hours, allowing one flight per fortnight. Early flights might take longer than this, but the objective should be achievable with experience. Operating in northern Europe, continuing flights at this rate might be as much interrupted by the weather as by operational problems.

An estimate has been made of the number of flight trials likely within the RADEM program. The program was divided up into three phases:

- Phase 1: Proving flights on the first vehicle (26 flights).
- Phase 2: Complete flight envelope testing to maximum speed, altitude, and range, including a second vehicle (33 flights).
- Phase 3: Advanced experimental flights, possibly including testing of airbreathing engine components (36 flights).

Past experience with the X-15 program achieved an average turnaround of 103 shift-hours per flight with a ground support crew of 350. The DC-X1 program had achieved 2,800 man-hours per flight with a crew of 55, but the vehicle was less demanding (fewer subsystems and there was no TPS surface inspection on the initial vehicle). The larger and more demanding SR-71 also achieved hundreds of maintenance man-hours per flight hour.

As a prototype, Radem could be expected to cost an order of magnitude less than an operational vehicle of the same size. Such a reduction was planned to be achieved by

- using an off-the-shelf engine;
- using an available TPS without further development;
- using specific rigs for critical areas, but not having any integrated ground testbed;
- using off-the-shelf equipment for all subsystems;
- choosing structural and fabrication methods on cost grounds;

- using experimental aircraft quality standards;
- leasing rather than buying the carrier aircraft.

It was hoped that significant cost and schedule reductions could be made by following an experimental aircraft approach. But the high performance demanded of Radem included experimental areas within the system that were relatively expensive because this project would be the first opportunity to merge aviation and rocketry experience in Europe. The same conclusions were drawn from comparable work in other European countries, specifically in Germany within the Sänger and its flight test vehicle approach.

Summarizing remarks

Final presentation of the results of Radem study work took place at ESTEC on June 30, 1994 in the presence of a large audience, including representatives from Russia and the Ukraine. The main conclusions drawn from the extensive international collaborative effort were summarized as

- an air-launched, rocket-powered demonstrator capable of proving key issues associated with future reduced operating cost of launch vehicles appeared feasible using available off-the-shelf technology;
- a vehicle such as this would be capable of demonstrating flight to Mach 12–15, as well as proving essential operating concepts;
- a vehicle that would be based around an existing LOx/LH_2 engine such as the HM-7B (ESA) or the KRD (Russia) (some performance gains might be possible by combining these engines with the RD-120 LOx/kerosene engine as an initial booster);
- substantial program cost reductions would be possible by using available equipment and a progressive integration and flight test development program;
- suitable carrier aircraft existed in the An-225 (Ukraine);
- a combination of ESA and CIS technology would be the most effective way of implementing such a program (a wide range of possibilities for work share existed).

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